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ADVANCED ION PROPULSION SYSTEMS FOR AFFORDABLE DEEP-SPACE MISSIONS

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ABSTRACT

A key feature of future deep-space science missions will be the need for significantly greater on-board propulsion capability. To meet this need, ion propulsion based on the system that flew on NASA's Deep Space 1 spacecraft has now entered the mainstream of propulsion options available to deep-space missions. The next most likely science mission to use ion propulsion is the comet nucleus sample return (CNSR) mission. CNSR has recently been identified by the Solar System Exploration Subcommittee as the highest priority new mission for NASA's Exploration of the Solar System theme. Ion propulsion for CNSR enables the use of a smaller, less expensive launch vehicle, and significantly shortens the overall trip time. A trade study for CNSR was performed to identify engine and system technology improvements, which provide the greatest mission benefits for the lowest additional risk. This trade study indicated that the maximum specific impulse of the ion engine should be increased from 3100 s to 3800 s and that the maximum engine input power should be increased from 2.3 kW to 3.2 kW. Simultaneously the engine total propellant throughput capability must be increased from the 80-kg NSTAR design point to approximately 180 kg. A focused technology program to make these advances is underway.

INTRODUCTION

After a development history spanning nearly forty years, the first use of solar electric propulsion (SEP) on a deep-space mission began with the launch of the Deep Space 1 (DS1) spacecraft on October 28, 1998. This event marks a major milestone in the development of advanced propulsion for deep-space missions. The DS1 spacecraft uses a single-engine ion propulsion system (IPS), provided by the NASA Solar electric propulsion Technology Applications Readiness (NSTAR) project, as the primary on-board propulsion system. This propulsion system is designed to deliver a total ΔV of 4.5 km/s to the 486-kg (initial wet mass) DS1 spacecraft while consuming only 81 kg of xenon.

Ion propulsion has now entered the mainstream of propulsion options available for deep-space missions. This is important because many of the deep-space missions that are relatively easy to perform from a propulsion standpoint, such as planetary flybys, have already been accomplished. Future high priority mission classes, which include sample return missions and outer planet orbiters, place substantially greater demands on the capabilities of on-board propulsion systems. Ion propulsion can help make these missions affordable and scientifically more attractive by enabling the use of smaller, lower-cost launch vehicles, and by reducing flight times.

Several scientifically interesting deep-space missions are now looking to the use of ion propulsion to significantly reduce total mission costs. These missions include Comet Nucleus Sample Return (CNSR), Venus Surface Sample Return (VSSR), Saturn Ring Observer, Titan

Explorer, Neptune Orbiter, Europa Lander, and various Mars Sample Return options. Because these missions are more difficult, from a propulsion standpoint, than those used to justify the development of the NSTAR IPS technology, they benefit significantly from improvements to the ion propulsion technology that flew on DS1. Typically, the greatest overall benefit comes from increasing the total impulse capability per engine. As the engine total impulse capability is increased, fewer engines are required for a given mission resulting in substantial savings in mass and cost. Additional savings may be obtained for some missions by increasing the maximum engine specific impulse, resulting in significant propellant mass savings.

This paper describes the results of a trade study, which was performed to identify the best ion propulsion technology and system architecture to be developed in support of the CNSR mission. All the technologies and system architectures considered in the trade study are derivatives of the single-engine, ion propulsion system developed by the NSTAR project for DS1.

NSTAR PROJECT

The NSTAR project was initiated in 1992 and was designed to overcome the barriers preventing the use of Solar Electric Propulsion (SEP) on deep-space missions. To accomplish this, the project had to achieve two major objectives:

1. Demonstrate that the NASA 30-cm diameter ion engine has sufficient life and total impulse capability to perform missions of near-term interest.
2. Demonstrate through a flight test that the ion propulsion system hardware and software could be flight qualified and successfully operated in space, and demonstrate control and navigation of an SEP-based spacecraft.

By all measures, these objectives have been met with unqualified success. Aside from an initial hiccup, the operation of the NSTAR ion propulsion system (IPS) on DS1 has been flawless, and it successfully provided the ΔV required for the July 29, 1999 flyby of the asteroid Braille. Consequently, ion propulsion is now a credible propulsion option for future deep-space missions. Complete details of how the NSTAR ion propulsion technology was validated for deep-space missions are given in the NSTAR Flight Validation Report [1], as well as in a shorter version of this report [2] and in Ref. [3].

NSTAR IPS Technical Description

A simplified block diagram of the four major components of the NSTAR IPS is given in Fig. 1. The ion thruster uses xenon propellant delivered by the Xenon Feed System (XFS) and is powered by the Power Processing Unit (PPU), which converts power from the solar array to the currents and voltages required by the engine. The XFS and PPU are controlled by the Digital

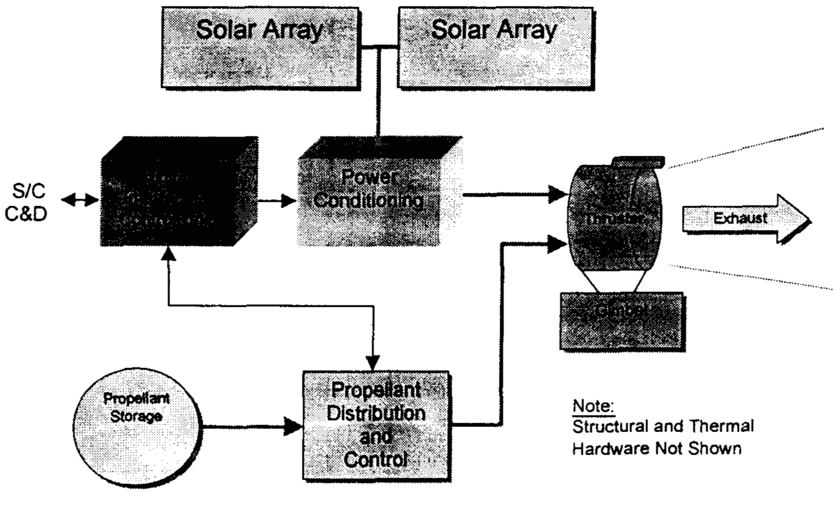


Fig. 1 Single-string ion propulsion system block diagram

Control and Interface Unit (DCIU), which accepts and executes high-level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system. To accommodate variations in the solar array output power with distance from the sun, the NSTAR IPS was designed to operate over an engine power range of 500 W to 2,300 W. Discrete levels within this range are referred to as "throttle levels". The mass of the NSTAR IPS as flow on DS1 is given in Table 1.

Table 1 NSTAR IPS Component Masses

Component	Mass (kg)
Ion Engine	8.33
Power Processing Unit (PPU)*	15.03
XFS minus Xenon Propellant Tank	12.81
Xenon Propellant Tank	7.66
Digital Control and Interface Unit (DCIU)	2.47
PPU to Ion Engine Cable	1.70
Total	48.00

* Includes 1.7 kg for micrometeoroid shielding

1.0 Ion Engine. The NSTAR 30-cm diameter flight ion engine was fabricated by Hughes Electron Dynamics (HED) and has four main components: the discharge chamber in which the xenon propellant gas is ionized; the ion accelerator system which extracts and accelerates the ions produced in the discharge chamber; the neutralizer cathode which injects electrons into the positive ion beam to provide space-charge and current neutralization; and the plasma screen which provides a grounded shield around the engine. The engine is based on technologies developed by NASA [4] and is designed to operate over an input power range of 500 W to 2,300 W, with a thrust of 20 mN to 92 mN and a specific impulse of 1950 seconds to 3100 seconds. The engine design life is 8,000 hours at full power, which corresponds to a propellant throughput capability of 83 kg of xenon and a total impulse capability of 2.65×10^6 Ns.

2.0 Xenon Feed System (XFS). The NSTAR xenon feed system is designed to store up to 81.5 kg of xenon propellant and provide three separate flow rates to the engine: main flow, cathode flow, and the neutralizer flow. The Xenon Control Assembly (XCA) was fabricated by Moog, Inc. and controls these flow rates to within $\pm 3\%$ over the range of 0.59 mg/s to 2.36 mg/s for the main flow, and 0.24 mg/s to 0.36 mg/s for the cathode and neutralizer flows. Xenon is stored as a supercritical fluid in a propellant tank (fabricated by Lincoln Composites, Inc.) which is maintained at a temperature between 20°C and 50°C.

3.0 Power Processing Unit (PPU). The PPU (fabricated by HED) is designed to take an 80-V to 160-V input directly from the solar array and supply the appropriate currents and voltages to start and operate the engine. This large input voltage range was designed to accommodate the expected variation in solar array output voltage resulting from a large variation in spacecraft-Sun distance for typical deep-space missions. During normal engine operation, the PPU provides four steady-state outputs: the beam voltage, the accelerator grid voltage, the discharge current, and the neutralizer keeper current which are provided by four power supplies. In addition, during engine startup the PPU provides heater power to the cathode and neutralizer heaters and an ignition voltage of 650 V to the cathode and neutralizer keeper electrodes. The power supply outputs are routed to internal relays, which allow them to be switched to one of two terminal blocks, so that a single PPU could be used to run either of two engines.

4.0 Digital Control and Interface Unit (DCIU). The DCIU (fabricated by Spectrum Astro, Inc.) serves as the data acquisition, control and communications unit in the IPS. The functions of the DCIU include: acquisition, storage and processing of the signals from the sensors on the XFS, and telemetry from the PPU; control of the valves in the XCA; control of the power supplies in the PPU; and communication with the spacecraft data and control system. The DCIU executes stored sequences that control IPS operating modes in response to high level commands generated on the ground or autonomously by the spacecraft. The communications with the PPU slice occur over an RS422 interface and commands from and telemetry to the spacecraft are transmitted on a MIL-STD-1553 interface.

NSTAR Flight Validation

One of the primary objectives of the flight validation activity is to verify that the system performs in space as it does on the ground. The parameters of interest to future mission planners are thrust and mass flow rate as a function of PPU input power. The variation in thrust as a function of PPU input power is given in Fig. 2 as measured early in the DS1 mission. These data indicate that the performance of the ion propulsion system in space agrees well with the expected performance based on ground-test measurements. The performance of the xenon feed system has been excellent. The mean value of the main flow and the two cathode flows are all within 1% of their respective planned flow rates [3].

In addition it is critically important to assess the extent to which the engine wear-out processes in space behave as they do in long-duration tests on the ground. Evaluation of the key electrical parameters that can be measured on DS1 which influence engine life suggest that the engine erosion rates should not be greater in space than they are on the ground. Indeed, these data [3] suggest that the ground test results are generally conservative. Engine wear affects the engine performance, and since we can't physically examine the thruster on DS1 to measure the wear rates, the next best thing is to map the engine performance after it has processed a significant amount of propellant. Fortunately, a unique opportunity will exist to do exactly this at the conclusion of what is now the DS1 science mission. Following the flyby of the Comet Borrelly in September 2001, the ion engine will have processed approximately 60 kg of xenon. This presents a unique opportunity to map the performance of an ion engine that has by far been the longest-ever operation in space. This opportunity will enable performance versus throughput comparisons with ground tests and will greatly improve our understanding of how well ground endurance tests reproduce actual in-space operation.

NSTAR Engine Service Life Validation

As mentioned above, one of the principal goals of the NSTAR program was to verify that the NSTAR ion engine has sufficient life to perform near-term, deep-space missions of interest. The NSTAR test program employed an extensive ground test activity together with the flight test on DS1 to validate the ion engine service life. Four long-duration ground tests of 1000, 2000, 8000 and 12000 hours were designed to identify unknown failure modes, characterize the parameters which drive known failure mechanisms and determine the effect of

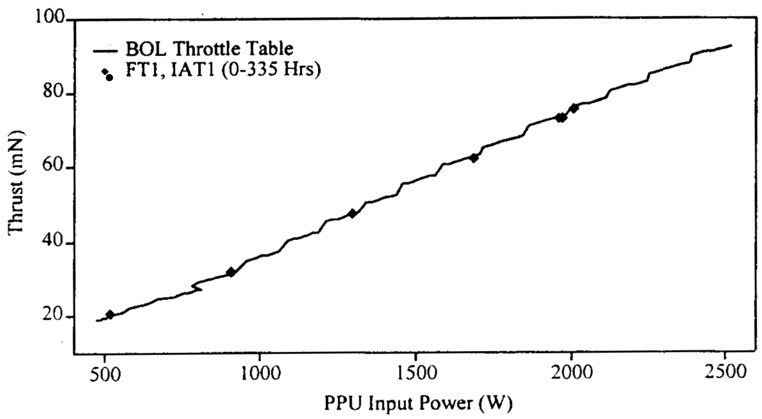


Fig. 2 Thrust measured in-flight as a function of PPU input power compared to the throttle table values.

engine wear on performance.

The 8000-hr test, which used an engineering model thruster fabricated by NASA GRC, was the most successful endurance test of a high-power ion engine ever performed (details of this test are given in [5,6]). A total of 8,192 hours of operation were achieved at the 2.3-kW full power point before it was voluntarily terminated. A total of 88 kg of xenon propellant was processed, demonstrating a total impulse of 2.73×10^6 N-s.

Inspection, measurements and analyses performed after the 8000-hr test indicated that none of the know failure modes were close to failing. In addition, only one new potential failure mode was identified. Assessment of this new failure mode indicated that it is a "soft" failure in the sense that even if it were to occur during a mission, it would not prevent normal operation of the thruster. The combination of detailed analyses and test data was presented to an independent review board which concluded that the NSTAR ion engine could process a total propellant throughput of 130 kg with a low wear-out failure risk with one caveat [7]. This caveat was that the average engine power level must be less than 2.1 kW. Operation at the full power point of 2.3 kW is allowed, but just not for the full 130-kg throughput. This restriction was imposed in order to obtain at least a factor of two margin on all known failure modes. Fortunately, most out-bound deep-space missions that would use solar electric ion propulsion tend to meet this requirement automatically since the available power decreases with increasing solar range.

COMET NUCLEUS SAMPLE RETURN (CNSR)

CNSR has recently been identified by the Solar System Exploration Subcommittee as the highest priority new mission for NASA's Exploration of the Solar System theme. This mission will return samples of volatiles and dust from the nucleus of a comet, and will provide new insight into our origins, evolution, and destiny. CNSR is targeting a launch in the 2005 to 2006 time frame, but launch opportunities to suitable comet targets occur nearly every year.

Advanced solar electric propulsion enables a total mission duration of 6 to 10 years, as well as the use of a much smaller, much less expensive launch vehicle. An illustration of the benefits enabled by ion propulsion can be obtained by comparison with the International Rosetta Mission being developed by ESA. The Rosetta spacecraft, which does not use ion propulsion, is being designed to rendezvous with the comet 46P/Wirtanen. The on-board, bi-propellant propulsion system will provide a total ΔV to the Rosetta spacecraft of 2.44 km/s and consume nearly 1600 kg of propellant. This results in approximately 55% of the spacecraft initial wet mass being propellant, with a final spacecraft dry mass of 1300 kg. Rosetta will be launched by an Ariane 5 and the total trip time to the comet is just over 9 years with no return to Earth.

In contrast, the CNSR ion propulsion system can deliver a 1300-kg spacecraft to the same target comet in approximately 2.6 years from a Delta IV-Medium launch vehicle. The ion propulsion system could then return the spacecraft (and comet samples) to Earth after an additional 4.5-year flight time, for a total mission duration of just over 7 years. To accomplish this the ion propulsion system must provide a total ΔV of about 10 km/s and will consume approximately 530 kg of propellant. Thus, the ion propulsion system enables *both* a much shorter trip time *and* a smaller launch vehicle than a spacecraft using a bi-propellant on-board propulsion system. Significantly, the ion propulsion system also includes sufficient propulsion capability to return the spacecraft to Earth!

CNSR Architecture Trade Study

The CNSR mission will likely be the first flagship science mission to use ion propulsion. The New Millennium, Deep Space 1 mission was first and foremost designed to demonstrate new technologies, with collection of science data only a secondary consideration. Science missions require that the bus subsystems, which include the on-board propulsion subsystem, have very high reliabilities. Specifically, CNSR will require that the ion propulsion system be single-fault tolerant.

A trade study was conducted to determine which engine and system technology improvements provided the greatest mission benefits for CNSR without introducing unacceptable technical risk. Five different engine technologies were considered:

1. The existing NSTAR/DS1 engine (I_{sp} = 3100 s, max. input power = 2.3 kW).
2. A low-Isp version of the NSTAR engine (I_{sp} = 3100 s, max. input power = 3.1 kW).
3. A medium-Isp version of the NSTAR engine (I_{sp} = 3400 s, max. input power = 3.1 kW).
4. A high-Isp version of the NSTAR engine (I_{sp} = 3800 s, max. input power = 3.1 kW).
5. A high-Isp, 5-kW derivative of the NSTAR engine (I_{sp} = 3800 s, max. input power = 4.6 kW).

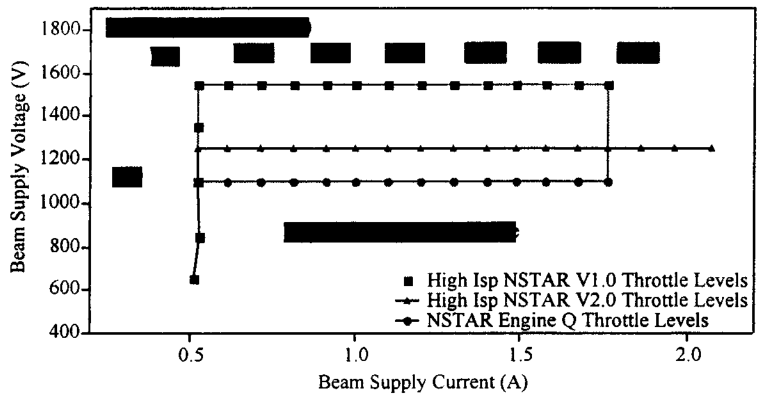
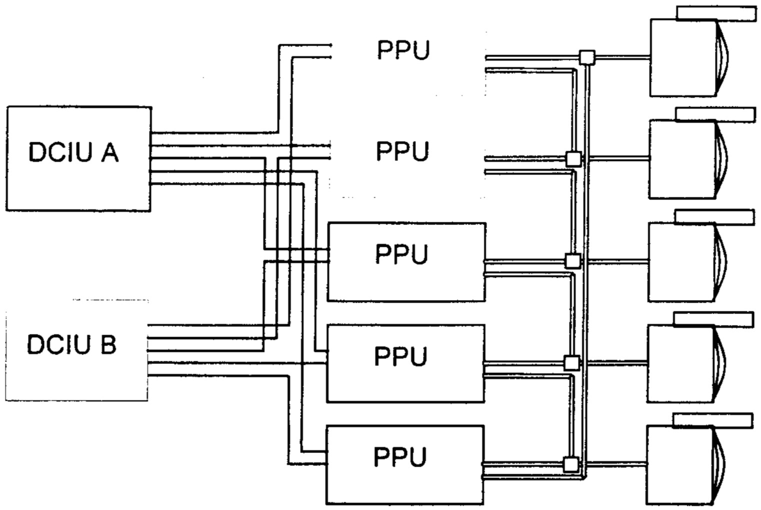


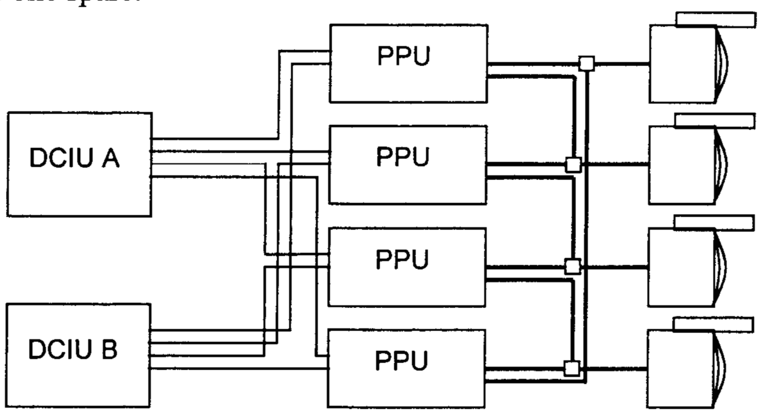
Fig. 3 Throttling envelopes for different engine technologies.

The throttling envelopes for these engines are given in Fig. 3. In addition, six different single-fault tolerant system configurations were considered:

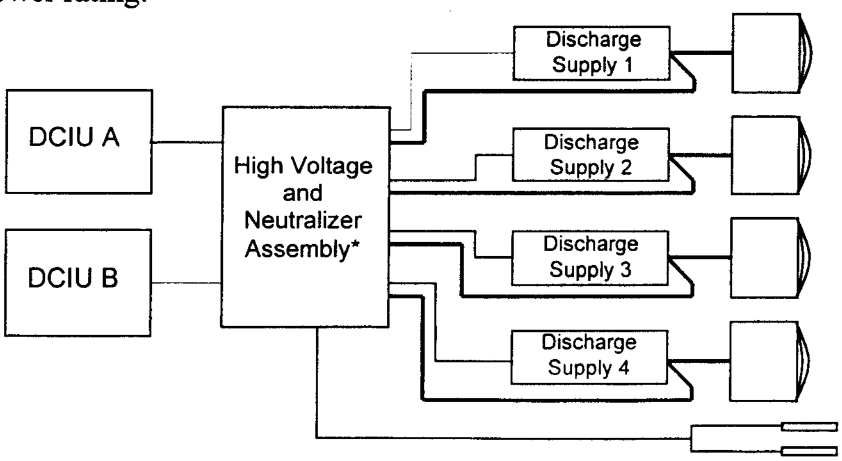
1. A conventional system architecture consisting of four engine-PPU strings identical to the single string which flew on DS1. A maximum of three engines is operated simultaneously. The fourth string is included to meet the single-fault tolerant requirement.
2. A conventional system architecture consisting of five engine-PPU strings identical to the single string which flew on DS1. A maximum of four engines is operated simultaneously. The fifth string is included to meet the single-fault tolerant requirement.
3. A conventional system architecture consisting of four engine-PPU strings with upgraded engines and PPU's. A maximum of four engines is operated simultaneously. Single-fault tolerance is obtained by operating the remaining three engine-PPU strings at 133% of their nominal power of 2.3 kW in the event of an engine or PPU failure early in the mission.
4. An unconventional system architecture consisting of four upgraded engines and a single, internally-redundant High-Voltage/Neutralizer Assembly (HVNA) which provides the high voltage and neutralizer power supply functions for all four engines, a central neutralizer cathode assembly, and four separate discharge power supply boxes. A maximum of four engines is operated simultaneously. Single-fault tolerance is obtained by operating the remaining three engines at 133% of their nominal power of 2.3 kW in the event of an engine or PPU failure early in the mission.



a) Conventional system architecture with a maximum of four engine-PPU strings operating simultaneously and one spare.



b) Conventional system architecture with a maximum of four engine-PPU strings operating simultaneously. If one engine fails the remaining three engine-PPU strings are operated at 133% of their nominal power rating.



c) Non-conventional architecture based on a single internally redundant HVNA and a central neutralizer subsystem.

Fig. 4 Ion propulsion system configuration options.

5. A conventional system architecture consisting of three engine-PPU strings with upgraded engines and PPU's. A maximum of three engines is operated simultaneously. Single-fault tolerance is obtained by operating the remaining two engine-PPU strings at 150% of their nominal power of 3.1 kW in the event of an engine or PPU failure early in the mission.
6. An unconventional system architecture consisting of three upgraded engines and a single, internally-redundant High-Voltage/Neutralizer Assembly (HVNA) which provides the high voltage and neutralizer power supply functions for all four engines, a central neutralizer cathode assembly, and four separate discharge power supply boxes. A maximum of three engines is operated simultaneously. Single-fault tolerance is obtained by operating the remaining two engines at 150% of their nominal power of 3.1 kW in the event of an engine or PPU failure early in the mission.

For each system it was assumed that the maximum input power available to the propulsion system is 10 kW. System configurations 2, 3 and 4 are shown in block-diagram form in Fig. 4.

Trade Study Results. The effect of higher maximum engine I_{sp} on the total propellant load is given in Fig. 5 as a function of the beginning of life (BOL) solar array power at 1 AU. These data indicate that increasing the maximum specific impulse from 3100 s to 3800 s reduces the required propellant mass by over 100 kg. Note, the solar array power levels given in this figure are substantially greater than 10 kW, which was assumed to be the maximum power available to the propulsion system. This is because the solar array is sized based on power required to depart from the comet and not for the beginning of the mission. Any power available above the 10 kW required by the ion propulsion system and the 450 W required by the spacecraft is not used.

For spacecraft designers and mission planners a key ion propulsion system performance parameter is the net spacecraft mass, which is defined as the spacecraft dry mass minus the dry mass of the ion propulsion system. The net spacecraft mass is given in Fig. 6 for selected combinations of engine and system technologies. The first number below each vertical bar represents the engine technology from the list above, and the second number represents the system technology from the above list of system configurations.

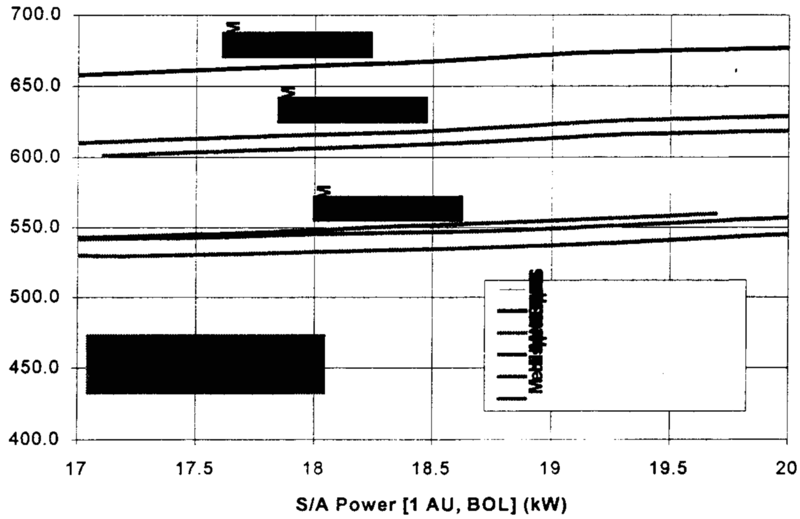


Fig. 5 Increasing the maximum engine Isp from 3100 s to 3800 s saves more than 100 kg or propellant for CNSR.

The total propellant throughput required per engine is given in Fig. 7. The numbers below each vertical bar have the same meanings as in Fig. 6. The data in Figs. 6 and 7 were used, in part, to down select to the (4,3) engine and system technology combination. This combination uses the high-I_{sp} NSTAR engine derivative which can be operated at up 3.1 kW and 3800 s along with an architecture consisting of four engine-PPU strings where each string is nominally operated at a maximum input power to the PPU of 2.5 kW. If there is a failure of an engine or PPU early in the mission the remaining engine-PPU strings are operated at 133% of this nominal power level (corresponding to an engine input power of 3.1 kW).

The (4,3) combination results in the best trade off between improved performance (i.e., increased net spacecraft mass) and low development risk. The data in Fig. 5 indicates that the 5-kW engine technology provides greater net spacecraft mass than the (4,3) combination. However, the development of a 5-kW engine, which is twice the power of the NSTAR engine was believed to be too risky within the timeframe of interest for CNSR. Similarly, the non-conventional system architectures (either 4 or 6 above) also enable increased net spacecraft mass, but this approach was also determined to be too risky and expensive to meet the needs of CNSR.

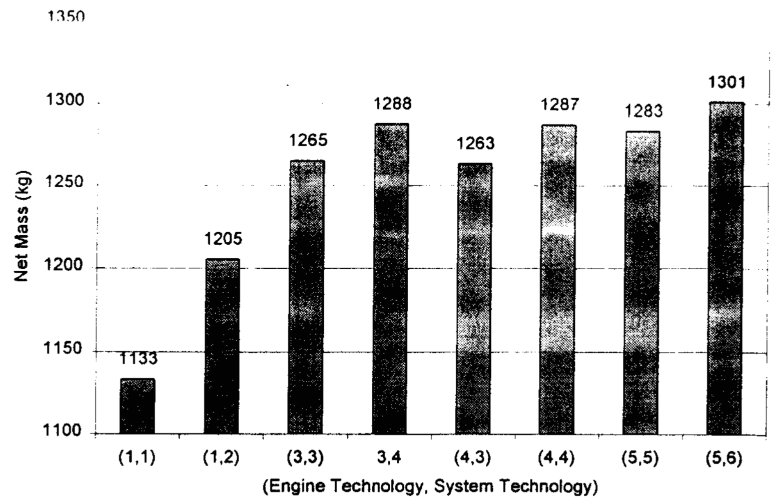


Fig. 6 Net spacecraft mass.

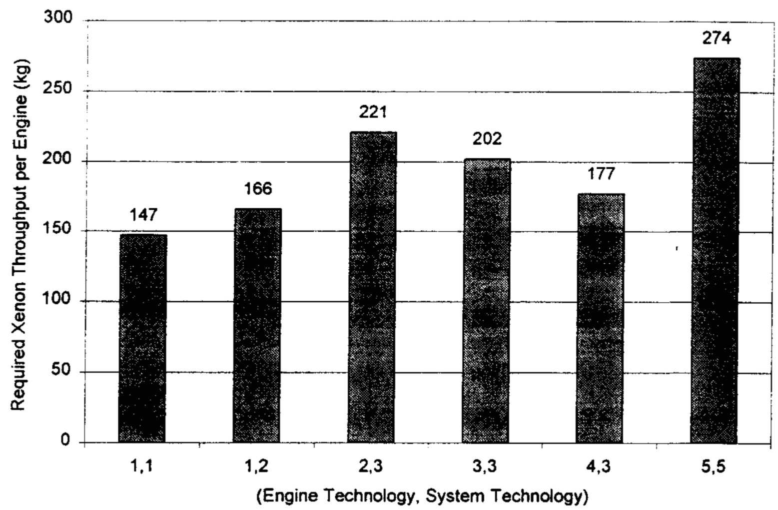


Fig. 7 Required xenon throughput per engine.

The selected combination (4,3) provides a significant net spacecraft mass benefit over the existing technology as represented by the (1,1) and (1,2) combinations, and the technology improvements required to obtain this benefit are straight forward and can be accomplished without invalidating the NSTAR flight and ground-test heritage.

CONCLUSIONS

Ion propulsion based on the NSTAR technology flown on Deep Space 1 is now a legitimate propulsion option for deep-space science missions. Future scientifically interesting planetary missions will place much greater demands on the capabilities of on-board propulsion systems than in the past. The use of ion propulsion systems can simultaneously help reduce the cost of planetary missions (by enabling the use of smaller launch vehicles) while improving the quality of these missions (by shortening the flight time). The comet nucleus sample return mission takes advantage of both of these benefits. For this mission an additional benefit in terms of increased non-propulsion-related spacecraft mass delivered to the comet can be obtained through the use of advanced NSTAR-derivative technologies. An ion propulsion system trade study was performed in support of the advanced study work for CNSR to determine the most fruitful technology investment areas. This study indicated that the best combination of improved performance and acceptable risk was obtained through the development of a high-I_{sp} derivative of the NSTAR engine in which the I_{sp} is increased from 3100 s to 3800 s and the maximum engine input power is increased from 2.3 kW to 3.1 kW. The corresponding ion propulsion system architecture makes use of the increased engine and PPU power handling capability to meet the single-fault tolerant requirement. Nominal full power operation of the system requires operation of all four engines at a PPU input power of 2.5 kW (which corresponds to an engine input power of 2.3 kW). If one engine fails at or near the beginning of the mission, the remaining three engine-PPU strings are operated at a PPU input power of 3.4 kW (3.1 kW into the engine). This approach eliminates the need to add an additional engine-PPU string to meet the single-fault tolerant requirement and results in significant mass, cost and volume savings.

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REFERENCES

1. Kakuda, R., "NSTAR Flight Validation Report," MOR-602, JPL D-18219, internal document, November 1, 1999.
2. Brophy, J. R., et al., "Ion Propulsion," internal JPL document, presented at the Deep Space 1 Technology Validation Symposium, February 2000.
3. Polk, J. E., et al., "In-Flight Performance of the NSTAR Ion Thruster Technology on the Deep Space One Mission," AIAA-99-2274, presented at the 35th AIAA/ASME/SAE/ASII Joint Propulsion Conference, June 21-23, 1999.
4. Sovey, J. S., et al., "Development of an Ion Thruster and Power Processor for New Millennium's Deep Space 1 Mission," AIAA-87-2776, presented at the 33rd Joint Propulsion Conference, Seattle, WA, July 1997.
5. J. E. Polk, et al., "The Effect of Engine Wear on Performance in the NSTAR 8,000-Hour Ion Engine Endurance Test," AIAA-97-3387, presented at the 33rd
6. J. E. Polk, et al., "An Overview of the Results from an 8,200-Hour Wear Test of the NSTAR Ion Thruster, AIAA-99-2446, presented at the 35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Los Angeles, CA, June 1999.
7. Parker, G., "NSTAR Risk Assessment for Discovery Proposals," Interoffice Memorandum PFPO-GLP-2000-002, April 5, 2000, internal JPL document.